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MASS BREAKDOWN OF THE SATURN V

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Abstract

The Saturn V launch vehicle is described, with emphasis on selected technical details. Mass data are presented graphically and discussed.

Introduction

In addition to its status as the largest successful launch vehicle of the twentieth century, the Saturn V is also unusual in that it was never used for commercial or military purposes. Therefore, detailed engineering information is in the public domain. The development history and many technical details have been recorded in extensive documents,[1-3] and very brief summaries exist as well.[4-5]

The purpose of this paper is twofold. One intent is to offer a moderate-length description of the Saturn V from the perspective of vehicle design and operation, with emphasis on particularly noteworthy features. Another is to present mass data which has not previously been available in a concise format. It is hoped that the information herein will be of value, both to history enthusiasts and to designers of future launch vehicles, including students.

A detailed mass breakdown of Apollo-Saturn number 511 (AS-511) has been condensed graphically herein.[6] This type of data is typically not publicized for launch vehicles, but the extremely high propellant fractions discussed herein are critically important. The Appendix speculates on why the importance of mass may be underappreciated.

The author was not a member of the Saturn V team, so the information presented is not firsthand. However, some technical insight is offered, and several opinions are expressed. This paper relies on the referenced documents and a few informal interviews. A best effort was made toward accuracy, considering that historical accounts do not always agree precisely. For example, the mass data and an event timeline[1] were used to check potentially imprecise descriptive information. Not everything could be included; notably absent is information about the engines, which constitute a complete subject in their own right.

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Saturn V Overview

The Saturn V could accelerate about 50 metric tons onto a translunar trajectory, in addition to the mass of the spent third stage. It was only launched 13 times, between 1967 and 1973, including a maiden flight beyond low earth orbit and no failures to reach orbit. Nine flights were translunar and 10 carried people, including the successful first translunar attempt. The success record is partly owed to extensive ground testing, not only of components but of complete stages.

The Apollo-Saturn stack is sketched to scale in Fig. 1. Labels on the left indicate the major components, with information about design and manufacture. Labels to the right are intended to inform the reader about further details including operation in flight. The left side arrows point to assembly joints, and those on the right indicate flight separations.

The first stage used dense propellants at a moderate specific impulse, while both upper stages benefited from a much higher specific impulse at a relatively poor propellant density. This design choice was easy to make. Avoiding high atmospheric drag requires the largest stage to be compact. A first stage is not lifted by other stages, so propellant mass and hence specific impulse are not critical. Upper stages must be lifted while full and they need not fly themselves through the atmosphere, so minimizing their propellant mass matters more than their volume.

The earlier Atlas-Centaur stack, as well as the Saturn I and IB vehicles, also used hydrocarbon in stage 1 and hydrogen in stage 2. More recently, the above principles have been carried further, with a trend toward solid rocket boosters lifting oxygen-hydrogen stages. Solid propellant is much denser than oxygen-hydrocarbon, at a still-lower specific impulse.

Figure 2 illustrates the mass breakdown of the entire vehicle, and shows operational data. Masses are indicated in metric tons in this paper. Mass bar graphs are adjusted to a standardized 100 percent scale, so that relative fractions can be readily interpreted. One noteworthy fact in Fig. 2 is that over half of the launch mass is liquid oxygen to be expended by the first stage! The S-IC is over 3/4 of the launch mass, and the first two stages together are 94%.

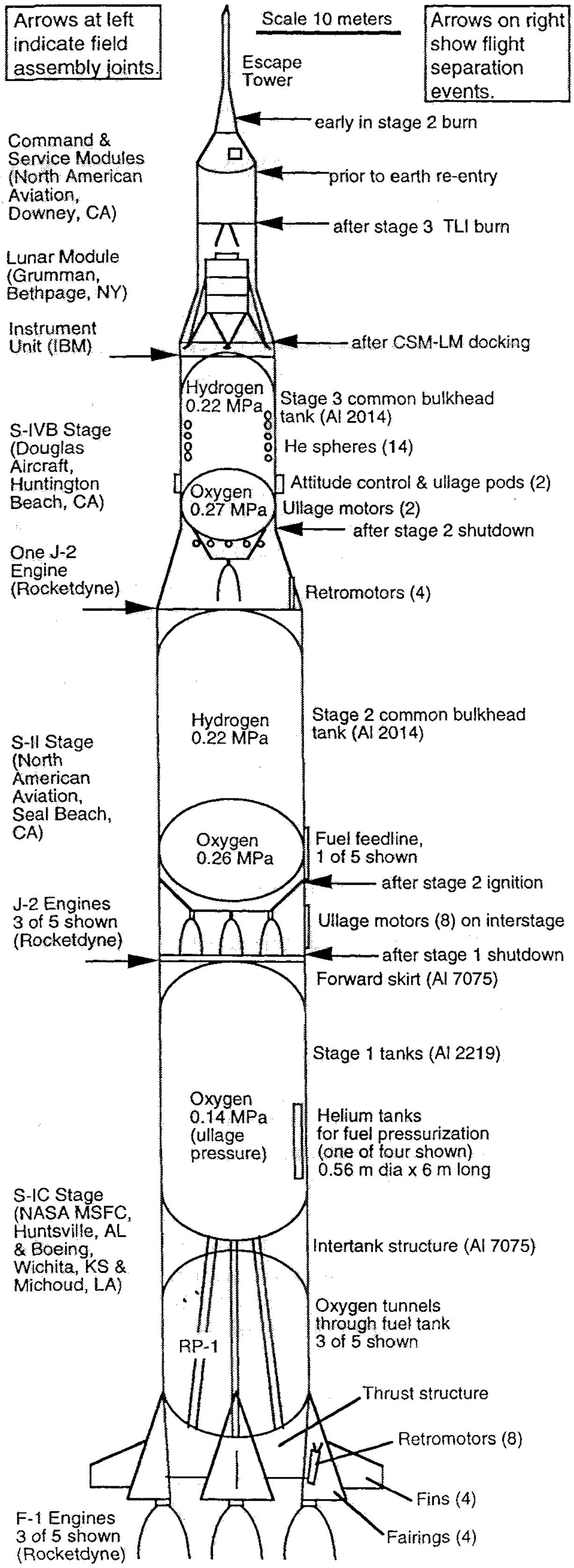

Figure 1. The Saturn V with Apollo spacecraft, to scale.

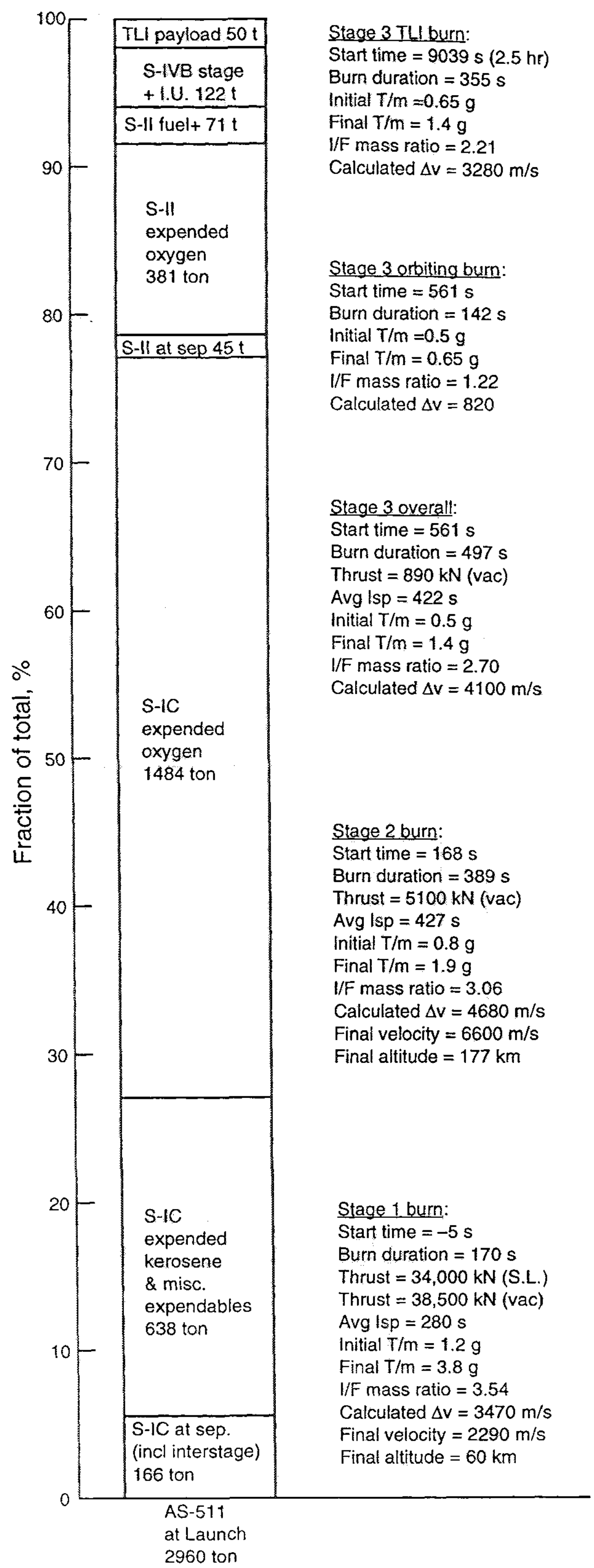

Figure 2. Top level mass & performance information.

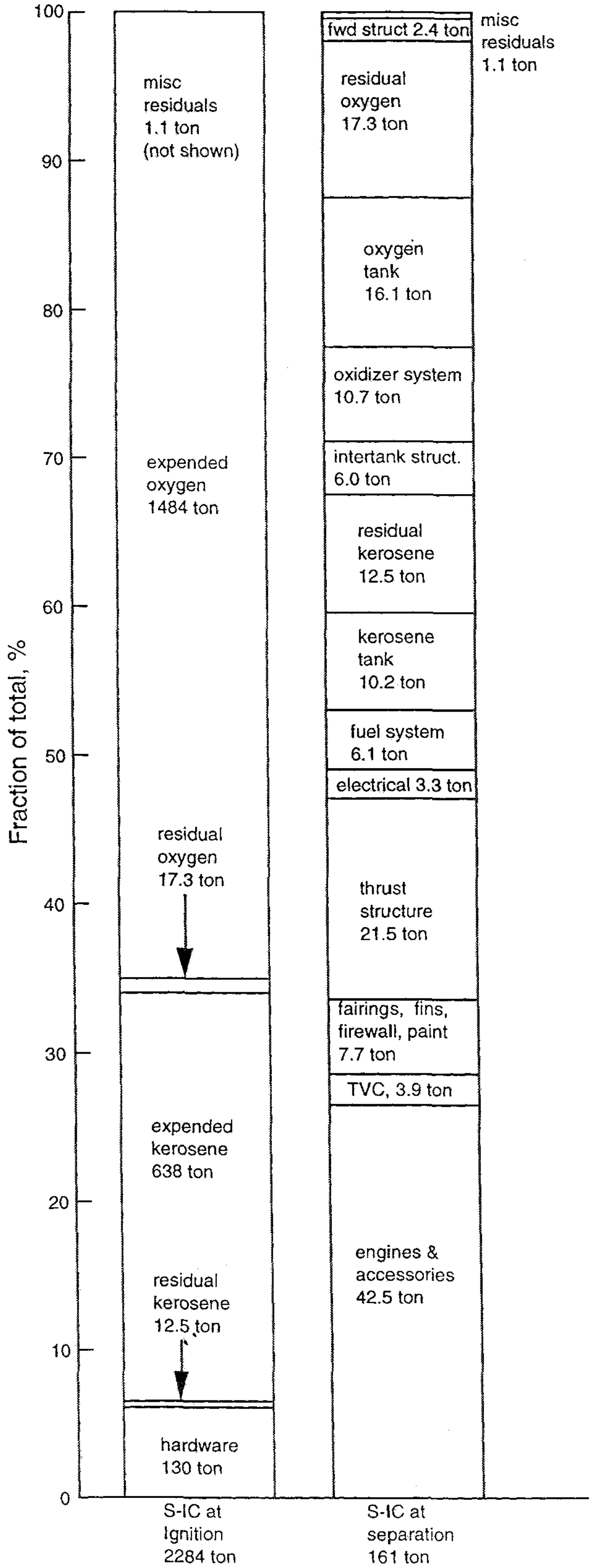

Figure 3. The S-IC stage is 93.0% useful propellant.

Stage 1

Figure 3 shows the mass breakdown of stage 1 both before and after operation. The order of items is representative of their vertical positions within the stage. At separation, the stage comprises 1/4 each engines and nontankage structure, 1/6 tankage, 1/5 residual fluids, 1/10 plumbing, and several percent control hardware.

The vast majority of the structural materials are aluminum alloys, with titanium used sparingly on the lower aerodynamic surfaces exposed to engine exhaust.[7] While the tanks are the largest parts of the S-IC, they are lighter than the nontankage structure sum. The thrust structure is relatively compact, but is much heavier than either tank.

As on the upper stages, the tanks are fabricated from aluminum plates, mostly milled away to leave membrane sections between stiffeners and thick edges for welding.[8] Hydrostatic testing is done at just 5% over the flight-rated pressures, in the range 0.14-0.19 MPa (plus liquid head). The skin thickness is tailored to local pressures, which vary with propellant head. The oxygen cylinder membrane thickness ranges from 6.6 mm down low, to 4.9 mm at the top. The fuel is less dense and its tank is not as tall, so the variation there is only 4.9 mm to 4.3 mm. The thinnest parts of the tank walls are less than half a thousandth of the 10.02 meter (396 in) diameter. The S-IC tank technology led to the Space Shuttle External Tank, which is fabricated at the same location, originally of the same 2219 alloy.

The S-IC oxygen is pressurized with 3.4 tons of its own vapor, and 0.24 ton of helium expels the fuel. In order to densify the stored helium and strengthen its aluminum pressure bottles, the 4 vessels are immersed in the liquid oxygen as shown in Fig. 1. The helium is loaded to 22 MPa at 5 hours before flight, just as the oxygen level reaches 98% full. In flight, the pressurant is heated by the F-1 engines on the way to the tank. The fuel system mass in Fig. 3 includes 2.2 tons of helium bottles.

As illustrated in Fig. 1, each engine receives oxygen through its own individual suction line. The five 0.43 m diameter ducts run inside 0.64 m tunnels installed through the fuel tank. The tunnels and ducts represent a large fraction of the oxygen system mass in Fig. 3, and most (11.5 t) of the residual oxygen fills these long ducts at separation.

In Florida's humidity, about 0.6 ton of frost accumulates on cold surfaces such as the uninsulated oxygen tank. While launch film shows ice falling off, half of it was estimated to remain at separation.[6]

For thrust vector control (TVC), the four outer engines are mounted on gimbals and each is steered by two hydraulic actuators, while the center engine is fixed. The same is true of stage 2, but a feature unique to stage 1 is that the hydraulic fluid is the fuel itself. Kerosene at 12.4 MPa from each fuel turbopump is routed to the hydraulic system for its own engine, then back to the fuel system. As a result, the control fluid supply is neither limited nor wasted, and a hydraulic failure on one engine cannot affect the steering of the others. At the pressure noted, each actuator exerts a 457 kN force. A flight set of the 1.5 m long actuators for the S-IC masses just over a ton.[9]

The S-IC program was an unusually close collaboration between government and industry, with the government maintaining significant in-house technical expertise. The stage design originated at the Marshall Space Flight Center, and whole stages were initially assembled and tested there.

Stage 1 separation

The inert mass of the S-IC differs by 5 tons between Figs. 2 and 3. As shown in Fig. 1, the aft end of stage 2 as manufactured includes a 0.6 ton piece which remains attached to stage 1, and a larger 4.4 ton interstage which separates independently. Both are excluded from the mass graphs of the individual stages in Figs. 3 and 4. However, this 5 tons of aluminum is effectively part of stage 1 hardware for determining vehicle performance, so it is included in Fig. 2 as such.

About 2.75 minutes after launch, the outboard F-1 engines are shut down when oxygen depletion is sensed, about 30 seconds after center engine cutoff. A half second later, the 8 ullage motors on the S-II aft interstage are ignited. Stage 1 is immediately separated and its 8 retromotors fire for 2/3 s at 3100 kN total, decelerating the S-IC at roughly 2 g.

The total thrust of the slower-burning ullage motors is 800 kN, which accelerates the remainder of the vehicle at just over 0.1 g to keep the liquids settled in the tanks. The stage 2 engines are ignited and their thrust ramps up within the 4.5 s burn time of the ullage motors. The 5 meter tall S-II aft interstage section with the spent ullage motors is separated tens of seconds later, at 3.25 minutes into the flight. The motor cases can be seen in launch film that shows this ring falling away.[10]

Up until this time, the escape tower at the top of the stack is available to pull the command module and astronauts away from the rest of the vehicle in the event of a malfunction. The crew jettisons it a few seconds after the aft interstage falls off. The escape tower and minor items, including numerical roundoff, account for a small discrepancy in the vehicle mass sum at the bottom of Fig. 2.

Stage 2

The left bar in Fig. 4 shows that the full S-II mass is over 3/4 oxygen. However, the oxidizer is only 1/4 of the total propellant volume. Therefore, locating the hydrogen above the oxygen (Fig. 1) keeps all propellant lines short, and reduces structural loading since most of the mass is down low. Five vacuum jacketed fuel feedlines (0.2 m diameter) are routed around the outside of the oxygen tank.

The absence of an intertank structure shortens the vehicle and reduces the mass of the expended second stage by perhaps 10%. The lower fuel bulkhead and the upper oxygen bulkhead are sandwiched together with about 0.1 m of insulation between them.[11] This thickness tapers away toward the periphery, as the aluminum vehicle structure must be continuous there to carry axial loads. While the term "common bulkhead" is used, there isn't a large reduction in bulkhead mass. In fact, the S-II bulkheads total 6 tons, almost half of the tank mass.

In addition to keeping the ~20 K hydrogen from boiling excessively while freezing a layer of oxygen, the insulation core in the common bulkhead tends to improve buckling strength, in the event that this bulkhead ever becomes compressively loaded. Normally, such a possibility is avoided by pressurizing the hydrogen to just 0.22 MPa, while the oxygen ullage pressure in flight is higher at 0.26 MPa. For the same reason, the oxygen is pressurized first, beginning at 3 minutes before launch, while the hydrogen pressure is brought up 1.5 minutes later. This pre-pressurization is done with helium for both tanks. Since the S-II begins operating within 3 minutes after launch, no additional pressurant is needed until its engines start. Propellant vapors from engine heat exchangers are used to maintain tank pressures thereafter. In Fig. 4, 60% of the residual oxygen is 1.9 tons of pressurization gas.

Despite the mass and packaging advantages of the common bulkhead, implementing it successfully on such a large scale was considered at the time to be risky.[1] This historical note is consistent with separate tanks not only in the S-IC stage but later in the Space Shuttle External Tank. A common bulkhead may limit inspection and can add complexity, because access is limited. For example, the S-II has internal plumbing in the oxygen tank, to reach the ullage for pressurization and venting.

On stage 2, the hydrogen tank is insulated externally. The tank wall thus operates at the lowest possible temperature, which improves the strength of the 2014 aluminum alloy. It is critical to exclude not only moisture from the insulation, but air as well since the latter's constituents

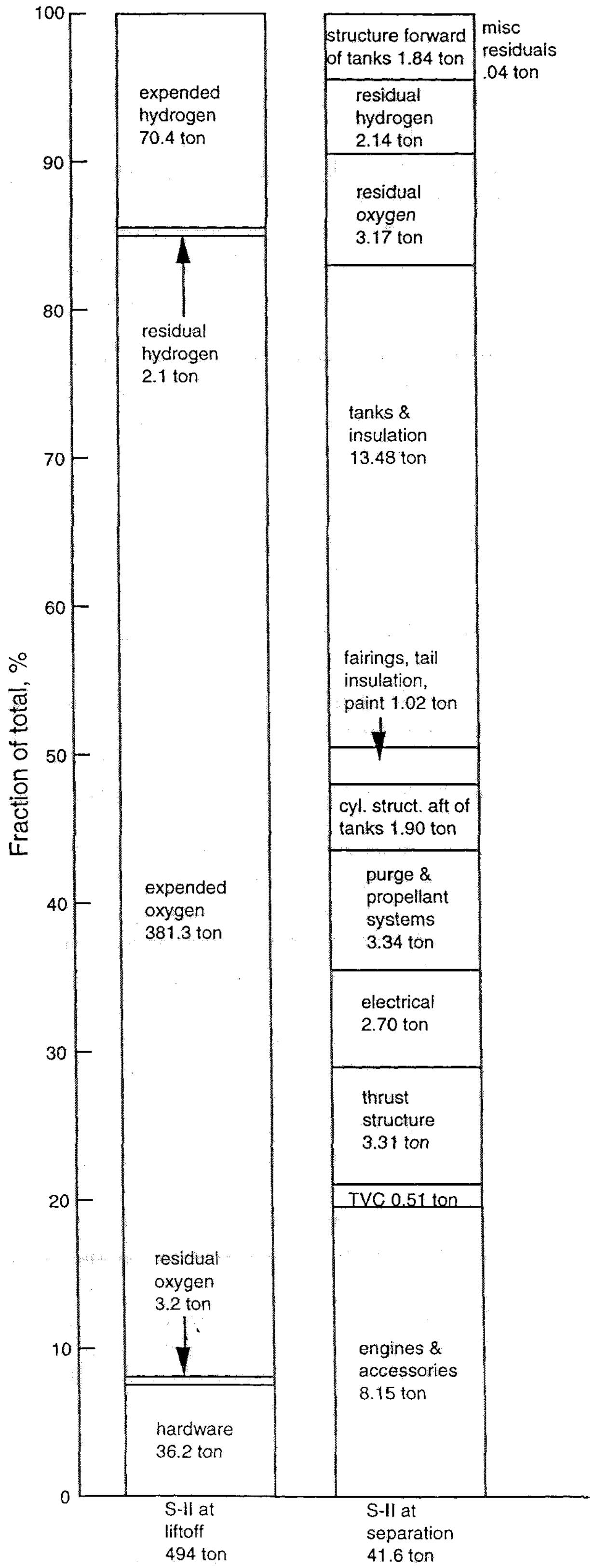

Figure 4. The S-II stage is 91.6% useful propellant.

solidify at the low temperature. Frozen air would both defeat the insulation and add inert mass. The 4 cm thick insulation layer was originally made up of honeycomb panels bonded to the tank, sealed with plastic film, and purged with helium. Due to difficulties, this gave way to spray-on foam insulation. Spray-on insulation was later adopted for the Space Shuttle ET.

Liquid hydrogen of course boils more readily than liquid oxygen, so timing is important as well as insulation in order to minimize losses. While all the liquid oxygen tanks on the Saturn V are filled between 5 and 7 hours prior to launch, the hydrogen tanks are filled between the 4.5 hr and 3.5 hr marks, just before astronaut boarding. In order to compensate for boiloff for all 5 cryogenic tanks, replenishment continued until just 3 minutes before launch. The tanks remained vented to the atmosphere during boiloff and replenishment, then were subsequently pressurized.

The S-II was the last of the 3 stages to be developed. The mere absence of a letter suffix indicates that it was not derived from an earlier version as the other stages were. Circa 1964, a mass reduction campaign to offset spacecraft growth was shouldered mostly by the S-II team, as it was too late to change the other stages. While understandable, it is ironic that the burden of "requirements creep" falls more readily to those having more work remaining. Schedule pressures and low structural margins may have contributed to the unintentional destruction of two ground test stages in 1965 and 1966. In the end, the S-II worked, and its propellant fraction of nearly 92% with a low density fuel is most impressive.

<u>Stage 2 separation</u>

Figure 2 indicates a greater mass than Fig. 4 does for this stage at separation. The 3.6 ton difference is the S-IVB's conical aft interstage, which becomes part of the S-II upon the latter's separation. For this reason, the historical mass list[6] treats the interstage separately, so it is omitted from Fig. 5 as well. The S-II retromotors are located on this conical structure. Their total thrust of 635 kN decelerates the spent S-II at 1.4 g.

Just as for stage 1 separation, the S-IVB solid ullage motors are ignited a fraction of a second before the S-II is dropped. At 30 kN thrust, the resulting forward acceleration of the S-IVB and spacecraft is 0.02 g for over 7 s. The ullage motors are mounted on retained structure, so the 2 spent motor cases are independently jettisoned several seconds after their burnout. Additional settling motors could be used for restarting. Instead, this function is incorporated into the liquid attitude control propulsion system, which permits multiple restarts.

Stage 3

Many features such as the common bulkhead are similar to those on the S-II, so this discussion focuses on unique aspects of the S-IVB. This uppermost stage of the Saturn V launch vehicle has only one engine, and it is designed to restart in orbit. Therefore, auxiliary propulsion is required for roll control during burns, attitude control during coasting, and propellant settling for one or possibly more restarts. The coast period of over 2 hours impacts the design of pressurization systems, as well as requiring extremely good insulation to minimize hydrogen boiloff.

As indicated in Fig. 2, less than a third of the burn time is needed to reach low earth orbit, and the rest is used later for translunar injection (TLI). In addition to relaxing the launch window, this operational scenario is invaluable for sending crews into deep space, as system tests can be performed before the final commitment to a long flight.

The auxiliary propulsion system (APS) consists of two modules, mounted on opposite sides of the aft skirt (Fig. 1). Each comprises an ullage thruster (320 N), a pitch thruster (654 N), and 2 thrusters for roll and yaw (654 N). These are pressure fed with 0.14 ton of hypergolic propellants stored in a pair of bladder tanks, pressurized by a helium bottle which is also in the APS module.[2]

Only 1/5 of the APS propellant is used up until separation. The vast majority remains available to maneuver the spent S-IVB to a solar orbit trajectory or a lunar collision, absent the convenience of dropping it in earth's atmosphere. The miscellaneous residuals in Fig. 5 include 0.24 ton of hypergols. The rest is mostly helium as explained below.

As on the S-II, the hydrogen tank is pressurized mostly by fuel vapor from the J-2 engine. However, the S-IVB oxygen is not pressurized by its own vapor. Note in Fig. 5 that the residual oxygen is proportionally tiny compared to Fig. 4. While potentially tons of oxygen gas would be needed due to ullage cooling during the long coast, the actual pressurant is only 0.31 ton of He, including that left in the source vessels. The oxygen tank is kept pressurized at 0.27 MPa by helium supplies not only before flight, but throughout ascent and during operation of the S-IVB's engine as well. Cooled helium is heavier, so oxygen tank pressurization is inactive during coasting. Prior to the TLI burn, the oxygen is repressurized with heated helium using a special oxygen-hydrogen burner.

Most of the helium is stored in spheres immersed in the liquid hydrogen, as indicated in Fig. 1. Extra helium is carried under the thrust structure. This gas is available in case of a burner failure or if a second J-2 restart is needed.

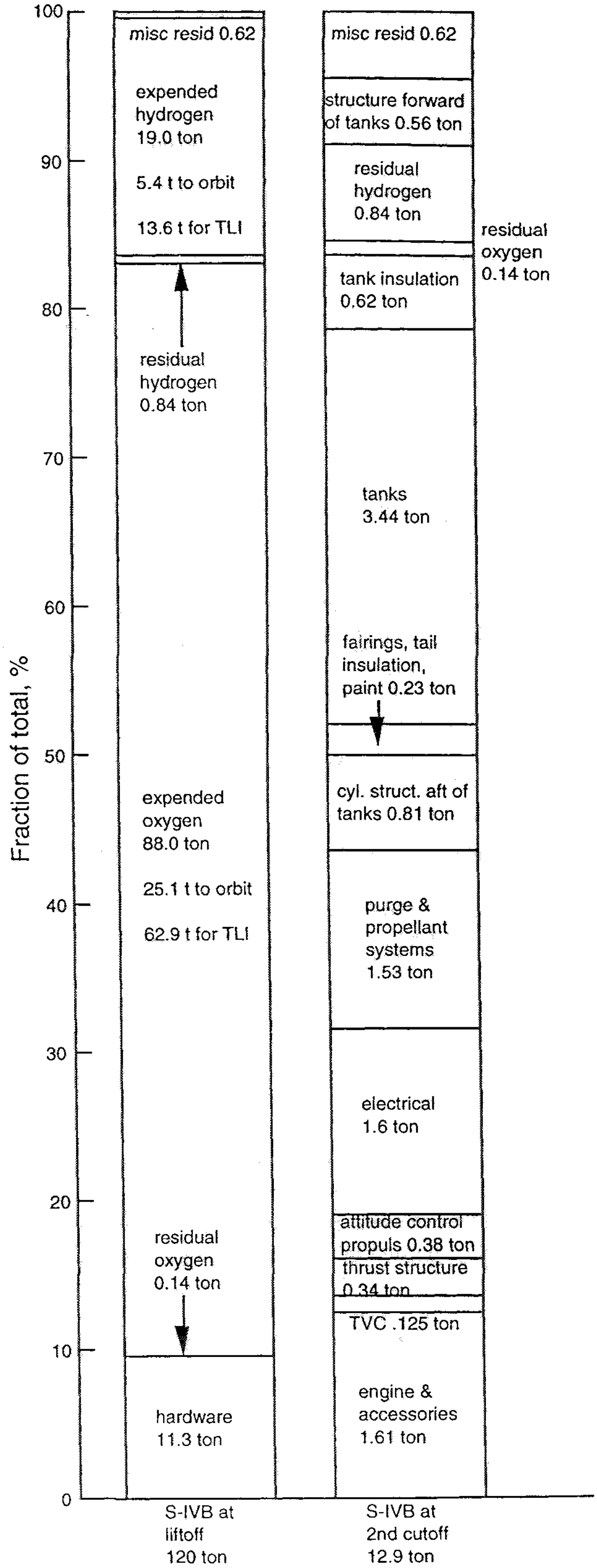

Figure 5. The S-IVB stage is 89.3% useful propellant.

The cryogenic propellants are kept settled in the S-IVB tanks most of the time. After the first J-2 burn ends, the APS ullage engines operate continuously for 1.5 minutes. During this period, the APS adjusts the orbit slightly and also rotates the vehicle to its desired orientation and rotation rate. This is one revolution per orbit so the astronauts' heads remain toward earth with the command module forward. Then at 1 minute past the J-2 cutoff, a propulsive vent connected to the hydrogen ullage is opened, to produce on the order of 100 N thrust during coasting. Two small hydrogen nozzles, balanced across the aft skirt, maintain settling to vent the boiloff without directly losing any liquid. About a ton of hydrogen is lost while awaiting the TLI, but the oxygen boiloff is only 24 kg owing to its much lower vapor pressure.

Nine minutes prior to the end of a nominal 2.5 hour coast, the hydrogen vent is closed. At this point helium from the system described above repressurizes the hydrogen tank. The APS ullage thrusters once again ensure that the tanks are settled by operating continuously for 1 minute before the J-2 is restarted. Restart requires other special subsystems, such as feedline recirculation for chilldown. Considering all the special subsystems (with redundancy) for orbital operations, restart, and deep space scuttling, the S-IVB's 89% propellant fraction is impressive when compared to that of the S-II.

Final separation of the spent S-IVB from the spacecraft does not occur until over an hour after the TLI burn is complete. During this time, the cryogenic tanks are vented and the APS is used by the astronauts for vehicle control while the command and service modules separate. The Apollo spacecraft re-orients itself and docks with the lunar module. Finally, approximately 4 hr after launch, the S-IVB separates from the re-connected spacecraft and the APS maneuvers it away about 6 hours post-launch.

The S-IVB has a high insulation fraction, almost 5% of the spent stage mass per Fig. 5. Other oxygen-hydrogen stages such as the S-II, Centaur, and the Shuttle ET have externally applied insulation. However, the S-IVB tank wall is insulated on the inside. Whereas the stiffeners and slosh baffles inside the stage 1 and stage 2 tanks run longitudinally and circumferentially, the "isogrid" waffle pattern is oriented at 45 degrees inside the S-IVB's 6.6 m diameter cylindrical wall.[12] An individual insulation tile is fit into each of the 0.24 m squares and overlaps the stiffeners to form a continuous layer. For each flight article, thousands of tiles were machined to fit and individually numbered so they could be set aside until after tank welding. The tiles were then installed by people inside the tank. Reference 1 describes the history of the insulation design choice and its development.

Other noteworthy history includes testing of completed stages at a Sacramento facility (adjacent to Aerojet) before shipment to the Florida launch site. This began with the S-IV which flew on the Saturn I, and was the largest launch vehicle tank at its time. In January 1964, a tank rupture due to overpressurization resulted in the largest-to-date oxygen-hydrogen fire that consumed an S-IV stage. Three years later, an S-IVB flight stage exploded on the test stand during a simulated launch sequence.[1]

In the 1950's, the Redstone Arsenal at Huntsville developed the Jupiter missile for the Army, while the Douglas team at Huntington Beach developed the Thor for the Air Force. The intense rivalry initially kept Douglas from planning to bid on the S-IV project. Max Hunter, then a young design guru at Douglas, tells this story. The Martin Company had been heavily promoting their Titan as an alternative to the Saturn I, which resulted in a loss of favor in Huntsville. Thus encouraged, Max accompanied his supervisor Elmer Wheaton on a trip to MSFC in 1960. Von Braun flatly stated that the Atlas (Convair) and Titan teams both had a better chance. At a lull in a conversation among his elders, Max asked, "Werner, considering that your Jupiter team is building the large S-I stage, I don't see how you can deny that the Thor team can do the smaller second stage" (for Saturn I). Von Braun's demeanor softened, and a Douglas engineering team was brought up to speed so they could submit a proper proposal. Their subsequent win was a shock to the industry.

<u>Comparisons among Saturn stages</u>

While the three stages have significant individual differences, it is interesting to compare their mass graphs (Figs. 3-5). Perhaps the simplest observation is that the fractional masses devoted to propellants, engines, tanks, etc. are similar over a wide range in stage mass, i.e. rocket stages can generally be scaled up and down. One item that has no reason to scale with stage mass is avionics. Note that the electrical system absolute masses are all nearly within a factor of 2 of each other, while the S-IC is over an order of magnitude heavier than the S-IVB.

The S-IC engines and structures are proportionally heavier than those of the upper stages, for several reasons. The lower specific impulse raises liquid mass and consequently the thrust load required for lifting. A higher vehicle thrust-to-mass ratio is required during the first stage burn, due to the nature of ascent trajectories out of gravity wells. Both of these facts influence structural mass as well as engine sizing. While all 3 stages need thrust structures to distribute the engine force around their tank perimeters, the one on stage 1 does more. The S-IC thrust structure has 4 posts which carry the loaded weight of the entire stack to the launch pad. Just before liftoff, the difference between thrust and weight is transmitted in the opposite direction to 4 holddown arms. The fins and aerodynamic engine skirts are also unique to the first stage. In contrast, the upper stages have essentially nothing aft of their tanks during operation, except the engines and minimal thrust structures.

Liquid density and stage propellant fractions

Figures 3-5 show that the total S-IC tank mass, which is directly affected by propellant density, is a smaller fraction of stage burnout mass when compared to the hydrogen stages. Considering that oxygen-kerosene is 3 times denser than the oxygen-hydrogen combination, the S-IC propellant fraction would be expected to exceed those of the upper stages. The propellant fractions are tallied in the captions to Figs. 3-5. Stage 1 does in fact have the highest percentage, even though its engines and structures are relatively heavier for reasons noted above.

To focus on the effect of propellant density, discount the intertank, fins, engine fairings, and half the thrust structure of the S-IC. Doing so reduces the spent stage mass by 24.5 tons, raising the propellant fraction to 94%. It is appropriate to also correct for significant differences in the lifting capabilities. The S-IC can lift 1.5 times its own loaded weight off the ground, while the same ratio is just 0.9 for the S-II assuming 4500 kN sea level thrust. If the masses of engines, TVC, and the thrust structure on the S-II are increased by the ratio 1.5/0.9, an extra 8 tons of inert mass reduces the S-II propellant fraction to 90%.

Considering the S-IVB, discounting the extra insulation and APS (with residuals), along with half the propellant system permits a fairer comparison to the S-II. In addition, the fractionally excessive electrical system should be scaled back 3 times. These changes reduce the inert mass by 2 tons, increasing the stage propellant fraction to 91%. However, the masses of the engine, TVC, and thrust structure should be multiplied by 2.25, using the same rule applied to the S-II above. An extra 2.5 tons for this thrust-to-mass adjustment reduces the propellant fraction back to 89%. This number is very close to the adjusted propellant fraction of the S-II (90%). Their agreement is consistent with expectations based on propellant density, and also suggests that stage sizing has a relatively minor effect on propellant fraction.

Tanks

Figure 6 shows tank mass details for all 3 stages. The labels along the horizontal axis indicate that the volume-to-mass ratio is similar for all of them, on the order of 100 m^3/ton. Note that the S-IC tanks are only twice as heavy

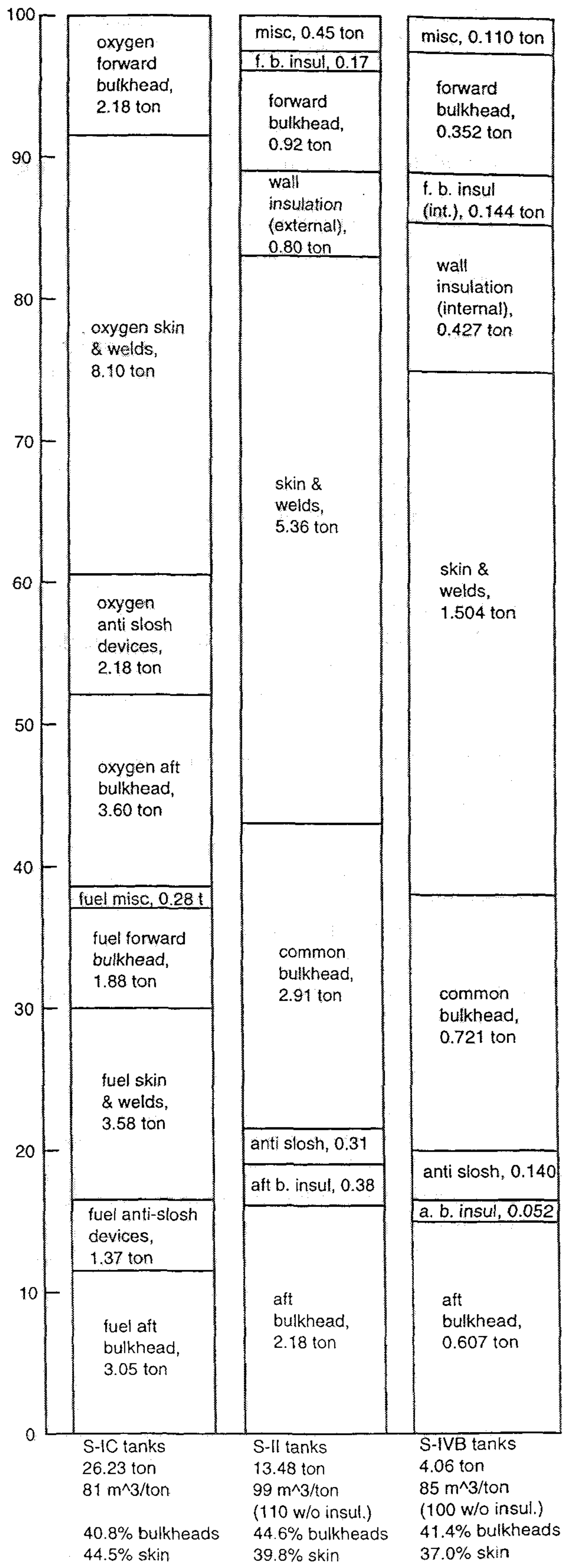

Figure 6. Tank mass breakdowns compared.

as those of the S-II, while the stage masses differ by a factor of 4. This is an effect of propellant density.

For consistency, Fig. 6 shows the tank parts in the order of their vertical locations. Note that the S-IC intertank is not included here, so that this graph compares the design of propellant containers without regard to the advantage of the common bulkhead design. However, if the 6 ton intertank structure (Fig. 3) is counted as part of the Stage 1 tank assembly, then the performance falls to 66 m^3/ton.

A wide range of insulation fractions is evident in Fig. 6, for reasons explained earlier. Another major difference shown in this graph is that anti-slosh devices account for over 1/8 of the S-IC tank mass, while their counterparts are only 2-3% of the other stages. If the first stage's atmospheric flight could be as smooth as that of the upper stages, a similarly small anti-slosh mass fraction would improve the S-IC tank performance to 91 m^3/ton. Even so, the largest tanks are still the least efficient ones in the absence of insulation. Other differences which might help to explain this include alloy choice, the high ambient operating temperature of the kerosene tank, and structural loads other than the ullage pressures noted in Fig. 1.

The fractional amounts of bulkheads and cylindrical wall skin are generally similar throughout Fig. 6. In all cases, both the total bulkhead mass and the total skin mass are each between 37% and 45% of the total tank mass (40-50% without insulation). Note that the common bulkheads represented in the rightmost 2 bars are about the same fraction of total tank mass as the 2 separate bulkheads in the middle of the S-IC stage. This observation is consistent with the fact that each common bulkhead is actually a pair together.

Within each stage, the forward bulkheads are lighter than the aft bulkheads because the latter ones carry propellant head in addition to the ullage pressures. For example, in the S-IC, a 17 m liquid oxygen depth exerts a whopping 0.23 MPa on the tank bottom at liftoff, which is a greater load than that of the gaseous pressurant! This alone explains why the S-IC ullage pressures are so low. In contrast, neither of the upper stages contains a dense propellant in a tall tank.

In Fig. 1, the S-II tank assembly appears to be essentially a larger version of the S-IVB tank, scaled up linearly in 3 dimensions. Thus it is not surprising that the rightmost 2 bars in Fig. 6 are quite similar and would be even more so with equal insulation fractions. However, 3-dimensional scaling (i.e. as mass) does not strictly apply to comparisons between the lower 2 stages, since they have identical diameters. The higher length-to-diameter ratio of the S-IC results in a lower bulkhead fraction (e.g. compare the lowermost bulkheads in Fig. 6) in spite of higher loading. Given equal diameters, it is not surprising that the uppermost bulkhead of stage 1 and the lowest bulkhead of stage 2 have the same absolute mass (2.18 t).

Additional structural insight

While a detailed analysis of Saturn V loads and structures is not the intent of this paper, some interesting information is provided in Table 1. For each stage considered in isolation, the thrust is compared to the stage mass at separation, and to the circular transverse tank area. The second row is generally consistent with proportionally heavier structures on stage 1. This is not to say that the upper stages are lightly loaded. They experience their highest loads when vehicle acceleration peaks at the end of the stage 1 burn.

The outstanding data point in Table 1 is at the bottom of the first column. The five F-1 engines push up on the S-IC with a force per unit area greater than the tank pressures! Thus the tank walls experience some axial compression during flight. While the S-IC oxygen ullage may not even need to be pressurized for the purpose of feeding the engines, it surely helps to avoid buckling.

Near the end of the first stage's burn, 80% of the remaining vehicle mass is above the S-IC, and over half of that is the S-II oxygen, per Fig. 2. During this time, most of the thrust is transmitted all the way through the S-IC and most of that stops at the S-II oxygen tank. Thus, both the stage 1 intertank and the stage 2 aft interstage would be expected to be rather heavy given that they carry this load and are not pressurized. Table 2 shows that this is true, by comparing the masses of cylindrical structures on the basis of their length. The overall trend is toward reduced structural mass higher in the vehicle, because the loads decrease also. In particular, the structures get significantly lighter above the second stage oxygen.

Table 1. Thrust levels compared to structures.

	Stage 1	Stage 2	Stage 3
thrust, MN	34 (S.L.)	5.1	0.89
$\frac{\text{thrust}}{\text{sep mass}}$, g	21.6	12.5	7.0
$\frac{\text{thrust}}{\text{tank area}}$, MPa	0.43	0.064	.026

Table 2. Comparison of nominally cylindrical structures.

structural element	mass/length, ton/meter	equiv. shell thick, mm
S-IVB forward skirt	0.18	3.1
S-IVB fuel tank wall	0.20	3.4
S-II forward skirt	0.50	5.7
S-II fuel tank wall	0.42	4.7
S-II aft interstage	0.72	8.1
S-IC ox tank wall	0.65	7.2
S-IC intertank	0.76	8.6
S-IC fuel tank wall	0.59	6.6

In the second column of Table 2, the equivalent cylindrical shell thickness is given. In all cases, the thickness is less than a thousandth of the diameter, which underscores the fact that launch vehicles must be exceedingly lightweight. If a modern aluminum beverage can wall could be scaled up, it would be 2 to 4 times thicker than the items in Table 2. Note, however, that these structures are far from being simple cylindrical shells. The tank skins with integral stiffeners have been mentioned previously. The intertank, interstage, and skirt structures have corrugated walls. In order to avoid buckling, they deviate much farther from being cylindrical shells than the tank walls do.

While high quality leaktight welds are critical for tank walls, welding is not an issue for the nontankage structures. Therefore, a stronger alloy having poor weldability (7075-T6 aluminum) is used for these items. Even so, Table 2 shows that the unpressurized structures are typically heavier per unit length than are nearby tank walls having an equal diameter. The sole exception is that the S-IVB forward skirt is slightly lighter than the tank wall, perhaps because its short length increases the buckling limit.

Conclusion

A half century ago, in the Spring of 1950, the Von Braun team was just settling in at Huntsville to continue rocket development begun over a decade previously. A generation of people has been inspired by the results.

It is an impressive success story, with at least 2 key ingredients that are absent from many aerospace programs today. One is that the technical work was shared between government and industry. As a result, government people maintained solid technical expertise which helped them manage contracts. Another is that extensive full-up testing was performed on complete stages, so that major failures occurred on the ground.

It is hoped that the data and insightful contributions herein will aid in understanding what was achieved, and how. The technical literature contains many individual papers on specific aspects of the Saturn V. A few are listed below for the benefit of readers having further interest.[13-21]

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Appendix

As one of two key parameters in the rocket equation, the mass change receives less notoriety than specific impulse. One possible explanation is that I_{SP} is entirely within the domain of rocketry, while mass ratio is a characteristic of complete vehicles, by nature more difficult for any individual or subsystem team to influence directly.

Another possible explanation is that the rocket equation is usually written as

$$\Delta v = I_{SP}\, g \ln \frac{m_I}{m_F} = I_{SP}\, g \ln (r)\,. \qquad (1)$$

The logarithm certainly tends to diminish the perceived importance of the initial-to-final mass ratio, r, relative to the importance of I_{SP}.

However, consider the propellant mass fraction, x:

$$\frac{\text{mass expended}}{\text{initial mass}} = x = \frac{m_I - m_F}{m_I} = 1 - \frac{1}{r}\,. \qquad (2)$$

Notice that $$r = \frac{1}{1 - x}\,, \qquad (3)$$

and consider the Taylor series expansion for the natural logarithm of Eq. (3). Equation (1) becomes

$$\Delta v = I_{SP}\, g \left(x + \frac{x^2}{2} + \frac{x^3}{3} + \frac{x^4}{4} + \frac{x^5}{5} + \ldots \right). \qquad (4)$$

For small propellant fractions, the first term dominates, and Eq. (4) is linear in both I_{SP} and x. As the propellant fraction increases, the higher order terms become important. Therefore, Δv increases more than linearly with x, but only linearly with I_{SP}. Equation (4) thus suggests that propellant fraction deserves as much attention as specific impulse.