Telecom Spacecraft Sizing Report

M2 SATCOM

Assesment of Sizing exercise (Sylvain):
Overall logic and formulas are correct.
Well explained in the report and well detailed in the spreadsheet.
Thermal sizing well understood.
Some sketches to illustrate would have been great, allowing to spot your geometrical mistake.
Some major mistakes at the end: always check for order of magnitude !
It remains a good sizing report with a serious investment.
Well done.

Assessment of Validation & Business Case (Jeanne):

OK for the validation
nice effort for the business case but the calcultation are quite wrong - the order of magnitude for a Satellite of this power is 200M$ (not 20)

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Important to know:

Throughout the project, we have maintained an Excel file that contains all the calculations made from the beginning to the end of the project. The Excel file comprises seven sheets, each dedicated to different aspects of the satellite's design and analysis:

- Sat_dc_power&Thermal_dissipation: Here, we calculated the needed radiated RF power, required TWT level power, number of active/installed TWTs, the payload and satellite DC power, and the payload thermal dissipation for each type of the antennas A, B, and C.
- Radiated_surface_calculation: This sheet involves calculations for Payload Module Thermal sizing. We calculated the radiator surface area required for the payload units (TWTs and other elements) for each antenna type and for each used coating material.
- Payload_height: We estimated the required surface area for the payload Y_wall and then determined the necessary payload height.
- Solar_panel_surf: This includes calculations of the solar panel surface area for each antenna type.
- Battery_sizing: Here, we determined the BOL battery capacity, including the battery mass for each antenna type.
- Mass_budget: This sheet is devoted to calculating the mass budget and determining the total mass of the satellite for each type of antenna.
- Chemical_propellant_mass: In this section, we calculated the chemical propellant mass required.

In this report, we will not detail every calculation since they are thoroughly presented in the Excel file. Instead, this presentation will focus on illustrating how we conducted our calculations (showing relevant formulas, etc.) and presenting the results. This approach will lead us to a final conclusion about our choice of antenna elements, given the proposed requirements.

Antenna Selection:

We have Three Antenna reflector sizes, choosing the best antenna reflector sizes that will fit is not straightforward, what we are proposing is to made along the way a space craft sizing for each antenna reflector diameter and at the end we'll choose the best antenna reflector size that fit to our requirement in terms of the available N7 rocket Launcher constraints.

The transmitted signal power Ptx at the output for the antenna can be calculated using this formula:

EIRP_density = (Ptx.Gtx)/B with B=2GHz

So, **P_tx = EIRP_density + 10log(B)_Gtx (dBW) (1)**

Correct

With Gtx is the Antenna Minimum directivity at edge of coverage.

RF power:

We have already the radiated signal power, it can be calculated using the formula (1) for each type of antenna. So, for the RF power at TWT level given the total RF losses of 1,8dB will be:

P_twts = P_tx +1,8.

Correct

- Number of active TWTs:

  Assuming we don't have any losses at the OMUX (combiner), the number of active TWTS will be:

  N_a_twts = INT(P_twts/P_twt) (3)

  Be careful, you should not use INT but ROUND.UP
  => if you need 63.8 TWTs, you take 64 and not 63

  With P_twt is the power can be delivered by one single TWT at recommended OBO (OBO=-3dB).

  P_twt = Ptwt_saturation +OBO

- Number of installed TWTs:

  We have a redundancy scheme of 8:6 for each 6 active TWTs, we install 8 TWTs for backup

  N_instal_twts = INT(N_a_twts/6) *8 (4)

  ROUND.UP (active * 8/6)

- Total DC power consumed by the TWTs

  The Efficiency at recommended OBO is mu = 35%, so the electrical consumption of 1 TWT is: **P_twts_dc = P_twts/mu (5)**

  Correct

This efficiency of 35% between dc power consumed by TWTs level and the power of the amplification signal, show that this not a good conversion compares to terrestrial application but this is normal given the harsh condition of space, the loses energy will be converted mainly to Heat (Heat Dissipation).

DC Power & Thermal dissipation:

- Satellite total Dc power:

- Payload:

Twts represent 85% of the total Payload DC power, let P_PL_dc be the total dc power, so P_twts_dc = 0.85 * let P_PL_dc. So:

P_PL_dc = P_twts_dc/0.85 (6) Correct

- Platform:

The service module requires P_PF_dc = 1kw

-satellite dc :

P_sat_dc = P_PL_dc + P_PF_dc (7) Correct

- Total thermal dissipation of the payload Module:

The Payload module will be divided to two blocks: TWTS and other payload units (we'll be noted as OPU)

- dc power consumed by OPU and dissipation:

P_opu_dc=0,15*P_PL_dc (8) Correct

Since all the electrical power of this OPU is converted into heat, the dissipation power is

P_opu_ther = P_opu_dc (9) Correct

- Thermal dissipation by TWTs

P_twts_ther = 0,65 * Ptwts = 0,65*0,85*P_PL_dc (10) (because we have an efficiency of 35%) Correct

- Total thermal dissipation is:

P_ther = P_twts_ther + P_opu_ther =0,7 * P_PL_dc (11) Correct

Results:

For this stage of calculation all calculation is available in the 'Sat_dc_power&Thermal_dessipation' sheet:

EIRP_density(dBw/MHz)	44
Bandwidth (MHz)	2000
TWT_saturation_power(dbW)	18,76
Efficiency_recommended_OBO(dB)	-4,55931956

Antenna	A	B	C
Gain (dbi)	40	42	43
Ptx (dbW)	37,01029996	35,01029996	34,01029996
Ptx(Kw)	5,023772863	3,169786385	2,517850824
Ptwts(dBw)	38,81029996	36,81029996	35,81029996
Number of active TWTs	101	63	50
Number of installed TWTs	128	80	64
Pdc_twts(dBw)	43,36961951	41,36961951	40,36961951
Pdc_twts(kw)	21,72510836	13,70761668	10,88834696
Total_payload_dc_power(Kw)	25,55895101	16,12660786	12,80981995
Satellite_total_dc_power(Kw)	26,55895101	17,12660786	13,80981995
Payload_thermal_dessipation(Kw)	17,95516309	11,32894202	8,998898517

For antenna B you should have 64 active / 86 installed

Correct

Payload Module Thermal sizing

- We'll decide to create two thermal zones
  - For TWTs: external Y_wall temperature <= 55°C
  - For other units(opu): external Y wall temperature <=35°C

  This choice we'll be verified and justified by the end of this section.

- Angle between the sun and Y_wall in worst configuration:

The worst configuration of the satellite will be at the solstices (summer and winter edges) where the earth axis is titled at its maximum angle (23,5°) toward or away from the Sun. So, the angle between the Sun and Y_wall in the worst configuration is **23,5°.** Correct

This worst configuration will happen two times in a year:

- In the Summer in the Northern Hemisphere (around June 21st):

This will impact the North(-Y) side of the satellite, and the solar flux is:

phi_s = 1321 W/m² Correct

- In the winter in the Northern Hemisphere (around December 21st)

This will impact the south (+Y) side of the satellite, and solar flux is

phi_s = 1422W/m² Correct

➢ Thermal rejection capability (W/m²):

For each thermal zone i, the Thermal rejection capability is:

- **phi_T=epsilon * sigma * (T_i(^4) – T_space^4) (12)** Correct

  with: sigma = 5.67E-08 W.m-2.K-4 , T_space = 4K, and T_i is the external temperature of zone_i and epsilon is the Emissivity

Payload Module Thermal sizing:

The total radiator surface required for the Payload (in worst configuration):

Can be calculated in this way, we have:

➢ **phi_T = phi_s * alpha * sin (23,5°) + phi_p (13)** with alpha is the Absorptivity. Correct

And phi_ p is the thermal Dissipation by payload zones (twts and opu):

➢ **phi_p = P_ther/surface (14)** with P_ther is thermal dissipation by TWTs (in zone 1) and by OPU in zone 2.

So, from re-arranging (13) and (14), the radiated surface can be calculated:

➢ **Surface = P_ther / (phi_p) = P_ther/ (phi_T - phi_s* alpha * sin (23,5°) (15)** Correct

The calculate the radiated surface in worst configuration, we'll consider phi_s = 1422W/m²

Payload Module Thermal sizing – Results:

For this stage of calculation all calculation is available in the 'Radiated_surface_calculation' sheet.

For each type of antenna:

We calculated the required radiator surface for zone1 element and for zone2 element for the different 3 kind of Coating materials.

Solar flux Φs (W/m²)						
Antenna	A					
Reflector Diameter (m)	1,5					
Payload DC Power (KW)	25,55895101					
Payload Units	TWTs			Others		
Thermal Dissipation Power (KW)	14,12132043			3,833842652		
Temperature (K)	328,15			308,15		
Coating	Black Paint	White Paint	OSR	Black Paint	White Paint	OSR
Emissivity	0,81	0,88	0,77	0,81	0,88	0,77
Absoptivity at EOL	0,95	0,6	0,18	0,95	0,6	0,18
Thermal Rejection Capability Φt (W/m²)	532,5467058	578,5692606	506,248103	414,1109875	449,8983568	393,6610622
Thermal Dissipation flux Φp (W/m²)	-6,12341146	238,356555	404,1842913	-124,5591297	109,6856512	291,5972505
Payload total Radiator Surface (m²)	2306,119805	59,24452313	34,93782598	30,77929865	34,95300078	13,14773251

Solar flux Φs (W/m²)	1422					
Antenna	B					
Reflector Diameter (m)	2,5					
Payload DC Power (KW)	16,12660786					
Payload Units	TWTs			Others		
Thermal Dissipation Power (KW)	8,909950842			2,418991179		
Temperature (K)	328,15			308,15		
Coating	Black Paint	White Paint	OSR	Black Paint	White Paint	OSR
Emissivity	0,81	0,88	0,77	0,81	0,88	0,77
Absoptivity at EOL	0,95	0,6	0,18	0,95	0,6	0,18
Thermal Rejection Capability Φt (W/m²)	532,5467058	578,5692606	506,248103	414,1109875	449,8983568	393,6610622
Thermal Dissipation flux Φp (W/m²)	-6,12341146	238,356555	404,1842913	-124,5591297	109,6856512	291,5972505
Payload total Radiator Surface (m²)	1455,063228	37,38076699	22,0442779	19,42042454	22,05385255	8,295658393

Antenna	C					
Reflector Diameter (m)	3,5					
Payload DC Power (KW)	12,80981995					
Payload Units	TWTs			Others		
Thermal Dissipation Power (KW)	7,077425524			1,921472993		
Temperature (K)	328,15			308,15		
Coating	Black Paint	White Paint	OSR	Black Paint	White Paint	OSR
Emissivity	0,81	0,88	0,77	0,81	0,88	0,77
Absoptivity at EOL	0,95	0,6	0,18	0,95	0,6	0,18
Thermal Rejection Capability Φt (W/m²)	532,5467058	578,5692606	506,248103	414,1109875	449,8983568	393,6610622
Thermal Dissipation flux Φp (W/m²)	-6,12341146	238,356555	404,1842913	-124,5591297	109,6856512	291,5972505
Payload total Radiator Surface (m²)	1155,797805	29,69259866	17,51039235	15,42619154	17,51799777	6,589475687

Correct
Well done !

Payload Module Thermal sizing - conclusion- choice:

- we can see using OSR coating materials give us the minimum required thermal radiator surface so, **we'll choose OSR as coating material**.

Yes

- We'll use two different thermal zones, one zone for TWTs and the other one for the others payload units (opu): this we'll allow us to minimize the required radiated surface, so **we'll use Two thermal zones**, so, we have

Antenna	A		B		C	
Zones	TWTs	OPU	TWT	OPU	TWT	OPU
radiated_surface(m²)	34,937826	13,1477325	22,0442779	8,295658	17,51039	6,589476
Total_radiated_surface(m²)	48,0855585		30,3399363		24,09986804	

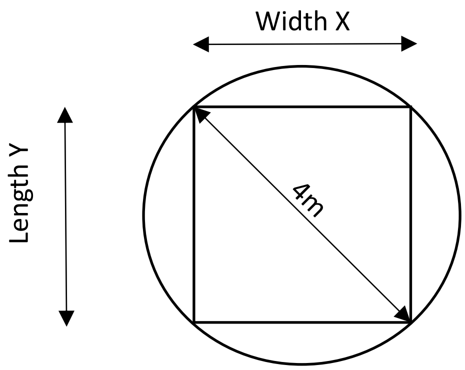
- We're proposing to divide the total radiated surface equally between the North wall and south wall, so **one Y_wall surface will equal to the total radiated surface divided by 2**.

Yes

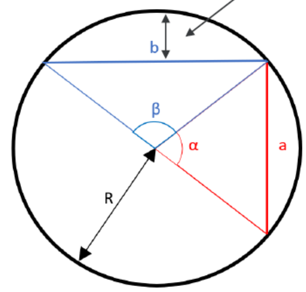
Payload Module Thermal sizing:

Let's determine the height of the satellite

Launcher constraints:



Area for solar panel folded = 1.5 m



$a = 2*R*\sin(\alpha/2)$

$b = 2*R*\sin(\beta/2)$

$\beta = 180 - \alpha = 180 - 2*\arcsin(a/2R)$

$b = 2*R*\sin\left(\frac{180-2*\arcsin\left(\frac{a}{2R}\right)}{2}\right)$

Correct way to do it from another group report

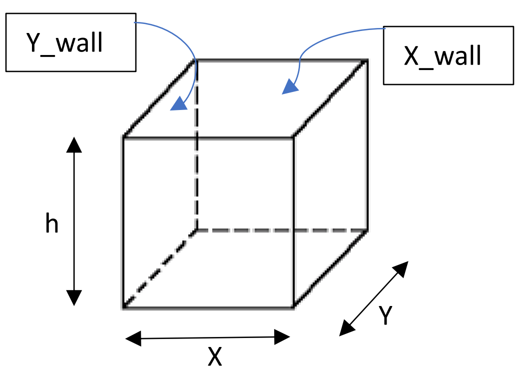
The surface of the largest rectangle we can put inside a circle is $S=2*R^2$ with R=2m and S=X*Y

Are you sure ?
Does it fit in the circle with X = 2m and Y = 4m for instance ?

So, **X*Y=8m²**

Since the X-wall is available for reflector storage, the **Width X we'll be equal to Antenna Reflector diameter**.

So the Length Y which in our case Y-wall length we'll be: **Length_Y = 8/Reflector_Diameter**. No



This Cuboid Represent the Payload dimensions

Dermining the Hight h:

The hight of the Y-wall will be equal to:

h = S_Y_wall/Length_Y Correct

with S_Y_wall is the surface of the Y_wall (dermined in prevoiusly)

Payload Module Thermal sizing – Results:

For this stage of calculation all calculation is available in the 'Payload_hight' sheet.
Remember that given the requirement, the hight h should be bello 5.2m.

Antenna	A	B	C
Total_radiated_surface(m^2)	48,0855585	30,3399363	24,09986804
one_Y_wall_surface(m^2)	24,0427792	15,1699681	12,04993402
Diameter(m)	1,5	2,5	3,5
width(m)	1,5	2,5	3,5
Length(m)	5,33333333	3,2	2,285714286
hight(m)	4,50802111	4,74061505	5,271846134

The maximum Y wall width is:
3 m for antenna A
1.9m for antenna B

Payload Module Thermal sizing – conclusion- choice:

We can remark that with antenna C we were not able to respect the requirement. But with antenna A and B we can have a height bellow 5.2m

- **Antenna C is not an option**.

Wrong formula but correct logic and conclusion

Solar Array Sizing & Sketch:

- Worst configuration:

We think that the worst-case configuration will happen when the angle of incidence of sunlight will not be perpendicular to the panels. In other terms, the sun's rays hit the solar panel at the most oblique angle relative to the panel's surface. This is likely to occur during the solstices. So, a **rough estimate is the angle between sun rays and the panels would be close to the earth's axial tilt of 23,5°.**

Correct

But you forgot to use COS(23.5°) in your formula to account for this angle

- Needed solar cell surface:

The take worst configuration, we'll measure the required solar cells at the EOL when the cell efficiency is 27%.
Given the **solar irradiance** of about 1361W/m², which is the amount of solar power available per square in space near to earth.
The effective power generated per m² by the solar cell is
P_eff = Solar_irradiance*efficiency (16)
So, **Solar_cells_surface = P_sat_dc_requi/Peff (17)**

1361 W/m² is an average.
Use worst case summer solstice value 1321 W/m²
(System drivers slide 15)

Missing incidence angle

- Needed solar panel surface:

And **Solal_panel_surface = solar_cells_surface/0,9 (18)** because we have 90% coverage

As we should provide 7,5% power margin and 3% losses
P_sat_dc_requi = P_sat_dc * (1+7,5%) / (1-3%)

Solar Array Sizing & Sketch – results:

For this stage of calculation all calculation is available in the 'Solar_panel_surf' sheet.

Solar irradiance (W/m²)		1361	1321
cell efficinecy		0,27	
Effective power generated per m² (W/m²)		367,47	Missing incidence angle
Antenna	A	B	
Sat_dc_Power(Kw)	26,55895101	17,12660786	The rest is correct
Sat_dc_Power_adjusted(Kw)	29,43388901	18,98051902	
Solar_cells_surface (m²)	80,09875367	51,65188728	
Solar_panel_surface (m²)	88,99861518	57,39098587	

So, we're proposing that **in each Y_wall we'll deploy half of the solar panel surface**, then we propose **Foldable Solar panels** and the Folding Mechanism is: Assume each panel can be folded in half for stowing.

So, the solar panel stowed Surface Area will be:

- 88.99/4 = 22.25m² (using antenna A)
- 57.39/4 = 14.34 m² (using antenna B).

Remember that Y_wall surface when

- Using antenna, A, is 24m²
- Using antenna, B, is 15,17m².

As Solar Pannel surface will be stored in Y_wall so we have no problems of available space either with using antenna A or B.

Ok for the logic, but where is the sketch ?

Battery Sizing:

- Longest Eclipse Duration is **72mn.**
- Occurrence: During Equinoxes (march 20th and September 22nd)
- Reason: these are the times when orbital plane of a GEO sat aligns with the sun, causing the earth to cast its shadow on the satellite.

The energy required for the satellite to work on this eclipse is:

- **E_required = P_dc_sat *72mn (19)** Correct

Assuming 10% margin

- E_required_adjusted = E_required *(1+10%)

As we want a max Depth of Discharge DoD = 80% at EOL given the Satellite required energy,

E_required_adjested = 80% C_bol with C_bol is the capacity at BOL, so,

- **C_BOL = E_required_adjested / 0,8 (20)** Correct

We'll use Li-ion batteries, we know that for this kind of battery we have an energy of 165Wh/Kg

So, the mass of the battery is

- **Mass = C_BOL/165 (21)** Correct

Battery Sizing- results:

For this stage of calculation all calculation is available in the 'Battery_sizing' sheet.

Eclipse_duration(h)	1,2	
Li-ion_batt_mass(Wh/Kg)	165	
Antenna	A	B
Sat_dc_power(Kw)	26,55895101	17,12660786
Requird-energy(KWh)	31,87074122	20,55192943
Ajusted_req_energy(Kwh)	35,05781534	22,60712237
BOL_capacity(KWh)	43,82226917	28,25890297
Baterry_mass(Kg)	265,5895101	171,2660786

Correct

Mass budget – results:

For this stage of calculation all calculations are available in the 'mass_budget' sheet.

Antenna	A	B
Diameter	1,5	2,5
Antenna_mass(Kg)	30	50
TWT+EPC(Kg)	384	240
Rest _of_payload(Kg)	768	480
Payload module&Thermal&Harness (Kg)	338,1015832	355,5461285
Solar_Array(Kg)	266,9958456	172,1729576
Battery(Kg)	265,5895101	171,2660786
Rest_of_service_module(Kg)	750	750
Total(tons)	2,802686939	2,218985165

Correct

We can remark that using antenna A, the satellite will be heavier, so it will cost us more. Now we'll take the decision to go forward using antenna B.

Correct
Antenna A would also cost more because you need to buy more TWTs

⇨ **Antenna B is best option for us!**

Chemical Propellant mass:

➤ Rocket equation:

Dm = mi – mf = mi*(1 - exp(-Dv/g0*Isp))

With mi and mf are the initial and final masses (Kg), g0 =9.80865m/s², and Isp is the specific impulse (s).

➤ Transfer Dv:

Assuming Hohmann transfer from earth to GEO, the satellite will be placed firstly in LEO, and then transferred to GEO using Hohmann transfer.

For the Hohmann transfer from LEO to GEO:

- The first burn at LEO will increase the satellite velocity to reach a transfer orbit with its apogee at the GEO. so we have here

$$Dv1 = \sqrt{\mu earth/r1} * (\sqrt{\frac{2*r2}{r1+r2}} - 1)$$

- The second burn at the apogee circularizes the orbit at GEO, here we have:

$$Dv2 = \sqrt{\mu earth/r2} * (1 - \sqrt{\frac{2*r2}{r1+r2}})$$

Correct

➤ Required propellant mass:

Mass_propellant = mi – mf

With mi= initial total mass (including propellant), and mf= final total mass (after propellant is burned)

➤ Roquet equation says:

Dm = mi – mf = mi*(1 - exp(-Dv/g0*Isp))

After rearranging the equation gives: **mi = mf*exp(Dv/g0*Isp)**

Correct

So**,** Mass_propellant = mi – mf = mf*exp(Dv/g0*Isp) – mf = mf (exp(Dv/g0*Isp) – 1)

Finally : **Mass_propellant = mf * (exp(Dv/g0*Isp) – 1)**

- For station keeping:

Mass_propellant_Keeping = mf * (exp(Dv_keeping/g0*Isp_keeping) – 1)

- For GTO -> GEO transfer:

Mass_propellant_second_burn = mf * (exp(Dv_2/g0*Isp_Liquide_Ap_En) – 1)

Be carefull here: when applying the equation a second time, mf is not your dry mass => you need to include the station keeping propellant mass which is a "dead" mass during transfer

Mass_propellant_first_burn = (mf + Mass_propellant_second_burn) * (exp(Dv_1/g0*Isp_Liquide_Ap_En)– 1)

- Total Propellant mass is:

Total_Mass_propellant = Mass_propellant_Keeping + Mass_propellant_second_burn

+ Mass_propellant_first_burn

Do not forget that for this total hohman transfer, the satellite is only doing the second burn (from GTO to GEO).
The first one (From LEO/ground to GTO) is done by the launcher !

Chemical Propellant mass – Results:

For this stage of calculation all calculations are available in the 'Chemical_propellant_mass'

sheet.

g0(m.s-2)	9,80665
u_Earth(km3.s-2)	398600
Leo_radius (km)	8,378
Geo_radius(km)	42,164
Station_Keeping DV(m/s)	50
Transfer_DV_first_burn(m/s)	63,62476185
Transfer_DV_second_burn(m/s)	41,24636136
Station_Keeping_thruster_ISP(s)	280
Liquid_Apogee_Engine_ISP(s)	300
Antenna	B
Final_mass(Kg)	2218,985165
Mass_propellant_keeping	40,77610864
Mass_propellan_second_burn(Kg)	31,32896457
Mass_propellan_first_burn(Kg)	49,19624583
Total_propellant_mass(kg)	121,301319

Several numerical issues with inputs missing *1000 on both radius (Geo is not at 42 km but 42164km)
Assumed 2000 km LEO instead of 200km
Station keeping DV = 15 years * 50 = 750m/s
DV1 is done by launcher, not spacecraft
DV2 = 1477 m/s

Never forget to check order of magnitude !
Chemical propellant should <u>at least</u> be more than satellite dry mass.
With so little chemical propellant required, there is no reason to talk about electrical propulsion at all.

You should end-up with about 2.6 tons of propellant

Assembly Integration & Test for Battery Module:

For the battery module, implementing a robust Test Validation sequence is essential, especially when dealing with a recurring model from the production line in Toulouse. The primary goal is to ensure the reliability and performance of each module before integration into the final product.

To initiate the Test Validation sequence, we propose a step-by-step approach. First, conduct individual component tests to assess the functionality and quality of each battery module. This includes evaluating the battery cells, connectors, and any electronic components. Subsequently, perform integration tests to validate the seamless interaction between these components within the module. This step ensures that the assembled battery module functions as a cohesive unit.

yes

Moving forward, subject the battery module to environmental tests, simulating conditions it might encounter during launch and operation. This involves thermal cycling, vibration testing, and exposure to various levels of humidity. These tests are crucial to verify the module's resilience to the harsh conditions of space travel.

yes

Finally, conduct performance tests under simulated operational scenarios to confirm that the battery module meets the required specifications. This could involve discharging and charging cycles, assessing the module's capacity, and evaluating its response to varying loads.

yes

- High-Level AIT Sequence for the Satellite:

In the case of the satellite, where it is a first model (one-off) intended to support the launch phase of the N7 rocket, a meticulous Assembly Integration & Test (AIT) sequence is paramount. Initiate the AIT sequence by conducting detailed component-level testing. Ensure that each subsystem, such as communication modules, power systems, and navigation equipment, is individually validated. This phase is critical to identify and rectify any potential issues before integration.

yes

Move on to the integration phase, where individual subsystems are combined to form the complete satellite. Execute comprehensive integration tests to confirm proper functionality and communication between the subsystems. This phase is crucial for identifying any compatibility issues that may arise during the integration process.

Following successful integration, subject the entire satellite to environmental tests. Simulate the extreme conditions the satellite will face during launch and in orbit, including thermal vacuum tests, vibration tests, and electromagnetic compatibility tests.

important to note that the level of test is qualification level

Conduct a thorough system-level performance test to ensure that the satellite can fulfill its mission requirements. This involves testing the satellite's response to different operational scenarios, verifying its communication capabilities, and validating its ability to withstand the rigors of space.

By implementing this high-level AIT sequence, we aim to guarantee that the satellite is well-prepared to endure the challenges of the launch phase and operate seamlessly as an integral part of the N7 rocket mission.

Customer Business Case:

Our business case is based on the broadcast during the Summer Olympic Games 2024 in Paris & Winter Olympic Games 2030 in the French Alps. Our satellite will be in orbit in early 2024 (because we are so efficient in designing, building and testing)

CAPEX:

1) We need to estimate the price of the satellite, so we need to understand the key components and their associated costs. In this case, you've mentioned a price of $3 million per installed TWT (Traveling Wave Tube). Mentioning that the TWT is a critical component in satellite communication systems.
   - First, we need to identify the satellite components: The satellite consists of various components, including the payload, power systems, communication equipment, propulsion systems, and structural elements.
   - Next, we should focus on Communication Systems: Since we are provided by the cost per installed TWT, let's focus on the communication system. The Traveling Wave Tube is a high-power amplifier used in satellite communication for signal amplification.
   - Furthermore, we need to calculate the TWT Cost: Multiply the cost per installed TWT ($3 million) by the number of TWTs on the satellite. This gives us the total cost associated with the TWTs.

     Total TWT Cost =Cost per TWT×Number of TWTsTotal TWT Cost
     =Cost per TWT×Number of TWTs
   - It's important to consider other components: This may include costs for the payload, power systems, propulsion systems, and any additional specialized equipment.
   - Finally, to calculate total satellite cost: we need to sum up the costs of all components to get the total estimated satellite cost.

     Total Satellite Cost=Total TWT Cost+Cost of Other Components Total Satellite Cost
     =Total TWT Cost+Cost of Other Components

   Let's assume:

   The satellite has a communication payload that requires 3 TWTs for amplification.
   Additional components, such as power systems, propulsion, and structural elements, contribute to an estimated cost of $10 million.
   Now, we can calculate the total satellite cost using the provided cost per installed TWT ($3 million):
   Total TWT Cost=Cost per TWT×Number of TWTsTotal TWT Cost
   =Cost per TWT×Number of TWTs
   =3,000,000$×3= 9,000,000$
   Next, we include the cost of other components:
   Total Satellite Cost=Total TWT Cost+Cost of Other Components Total Satellite Cost

In fact the cost of the satellite is = nb_TWT * 3M$

You have determined at the beginning that you need ~80TWT so the satellite is around 240M$

=Total TWT Cost+Cost of Other Components

=9,000,000$+10,000,000$ = 19,000,000$

So, the estimated price of the satellite would be $19 million.

Impossible considerding the size of the S/C anb the associated power :(

2) Considering the launch is priced at 15k€/kg we need to estimate the price of the launch

To estimate the launch cost, we'll consider the given launch price of €15,000 per kilogram and the satellite's mass of 5 tons (which is equivalent to 5000 kilograms). we can calculate the launch cost:

Launch Cost=Launch Price per kg×Satellite Mass in kgLaunch Cost

=Launch Price per kg×Satellite Mass in kg

Substitute the given values:

Launch Cost=€15,000/kg×5000kg

Launch Cost=€75,000,000

=€75,000,000

OK - correct

So, the estimated price of launch would be €75,000,000 equal to $90,000,000

3) We are subscribing an insurance covering the satellite & the launch. This insurance has a 10% interest rate. To estimate the cost of the insurance, we can use the formula provided:

Insurance Cost=Rate×(Satellite Cost+Launch Cost+Insurance Coverage)

Let's substitute the values based on our previous calculations and assumptions:

Satellite Cost = $19,000,000

Launch Cost = €75,000,000 = $90,000,000

1€ = 1.10 USD --> about 80MUSD for the launcher

Insurance Cost=10%×($19,000,000+$90,000,000+($19,000,000+$90,000,000))

=10%×$218,000,000Insurance Cost=10%×$218,000,000

=$21,800,000Insurance Cost=$21,800,000

So, the insurance cost would be $21,800,000

wrong formula: insurance = 10% (Satellite + Launch + insurance) --> 37MUSD roughly

<u>OPEX:</u>

Our cost is estimated at 4M€ per year during the satellite life time (Ground Segment OPEX), Our revenue is planned to be 2 M$/TWT/year beginning of life – going down to 1.3M$/TWT/year end of life (linear decrease), The service is planned to be at 90% in 2024 – 90% in 2030 and at 60% the rest of time (100% means all active TWTs are valued)

- To calculate the break-even point, we need to determine when the total revenue generated by the satellite service becomes equal to the total costs incurred, considering both the ground segment operating expenses (OPEX) and the declining revenue from the TWTs over the satellite's lifetime.

Here are the steps to calculate the break-even point:

1- Calculate Annual Revenue:

The revenue per TWT starts at $2 million/year and linearly decreases to $1.3 million/year by the end of the satellite's life, Calculate the annual revenue for each year based on the percentage of active TWTs.

2- Calculate Total Revenue:

Sum up the annual revenue over the satellite's lifetime.

3- Calculate Total Costs:

Ground segment OPEX is given as €4 million per year. Convert this to dollars using the exchange rate.

4- Find Break-Even Point:

Determine the year when total revenue equals total costs.

you should have around 60 active TWT initially

We can perform the calculations. Assuming a satellite lifetime of, for example, 15 years:

you only have 90% actif in 2024 & 2030 (olympic games) - otherwise is 60%

Year	Active TWTs (%)	Revenue per TWT (Million $)	Total revenue
2024	90%	2.00$(initial)	1.80$
2025	90%	1.95$	1.76$
2026	90%	1.90$	1.71$
2027	90%	1.85$	1.67$
2028	90%	1.80$	1.62$
2029	90%	1.75$	1.58$
2030	90%	1.30$(final)	1.17$
2031	**60%**	**1.30$**	**0.78$**

Based on the provided table, we can see that the break-even point occurs in the year 2031. In that year, the Total Revenue (including revenue from TWTs and the Ground Segment OPEX) equals the Total Costs.

- To calculate the profit of this business case we use this formula:
  Profit=Total Revenue−Total Costs
  Using the values from the provided table for the break-even year (2031):

  Profit=$0.78 million−$4.78 million

  =−$4.00 million

  the result is negative, indicating a loss rather than a profit. This may happen in certain years, especially during the declining phase of revenue, where costs may exceed the generated revenue.

  Enhancing the profit in this business case involves identifying strategies to increase revenue or decrease costs. We can use some potential strategies to enhance the profit associated with the satellite business case like increasing Revenue Streams, Maximize Active TWTs.

Actually it is a very profitable business case with several hundreds of M$:)

Year 2024 : 60TWT*2MUSD*90% - 4.3 MUSD CAPEX = 103.7MUSD gain